AE 6020 High-Speed Flow
Computer Project #1 (worth 10% of grade)
Due Date: Tuesday, February 24, 2004

On this assignment you will conduct Transonic Small Disturbance and Euler Eqn. analysis for two sets of airfoils.

First, download the TSD code from the course web site. This code is based on the Murman-Cole approach and uses a Line Gauss Seidel iterative procedure. Its boundary conditions are valid for subsonic freestream flow around a body with zero lift. The TSD code is designed to compute the flow over the upper half of a circular arc airfoil with a user specified thickness ratio.

Next, obtain the NASCART-GT code. Please complete and document the following tasks:

1. Using the TSD code, calculate the inviscid flow around the top half of a circular arc airfoil with a thickness ratio of 0.08 for M_∞ of 0.6, 0.7, 0.8, 0.85, 0.9, and 0.95 for $\alpha = 0^o$. Generate a single plot which shows all predicted Cp distribution for this code. Repeat this inviscid analysis and generate a Cp plot using the NASCART-GT code instead of the TSD code. Since the airfoil geometry is symmetric, set jsymin=1 in the NASCART-GT input file to compute only the top half of the airfoil.
2. Modify the TSD code so that it calculates a NACA 0012 airfoil (instead of a circular-arc airfoil). Note that you really only need to make changes to the "bcmat" subroutine in the code. Since the leading edge of the NACA airfoil is blunt, the slope of the airfoil is infinite at x=0. This will cause a numerical problem in the transonic SMALL DISTURBANCE solver. To remedy this problem, use the following for the airfoil slope:

$$\frac{dy}{dx} = \begin{cases} \dfrac{dy}{dx}(x) & for \quad x \geq 0.01 \\ \dfrac{dy}{dx}(x = 0.01) & for \quad x < 0.01 \end{cases}$$

In your report, describe the changes made to the program. Email an electronic version o your modified code to: stephen.ruffin@aerospace.gatech.edu

3. Repeat the analysis and plots done in part 1 for the NACA0012 airfoil (instead of the circular arc airfoil)
4. Using the NASCART-GT results, generate a single c_d vs M_∞ plot which includes both the circular arc airfoil and the NACA0012 airfoil results. Be sure to show the combined c_d for both sides of this symmetric airfoil and, as usual, normalize the drag coefficient by the airfoil chord length. With NASCART-GT, also generate and display (on the same plot) a c_d vs M_∞ curve for viscous flow (ivisc=2) at sea level atmospheric conditions using the adiabatic wall boundary condition. For the c_d vs M_∞ results, discuss the trends observed.

5. For all the Cp distributions, label and discuss important features of these plots and their change with flight Mach number. Also discuss differences in the results for the two airfoils. Compare the critical Mach numbers for the two airfoils.

Note:

The thickness distribution of a modified NACA 00xx series airfoil is given by

$$y(x) = \pm 5t\left[0.2969\sqrt{x_{\text{int}}x} - 0.126x_{\text{int}}x - 0.3516(x_{\text{int}}x)^2 + 0.2843(x_{\text{int}}x)^3 - 0.1015(x_{\text{int}}x)^4\right]$$

where the "+" sign is used for the upper half of the airfoil, the "-" sign is used for the lower half and $x_{\text{int}} = 1.008930411365$. Note that in the expression above x, y, and t represent values which have been normalized by the airfoil chord. In this problem we will consider a thickness of t=0.12.